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RANGER
RANGE SAFETY ANALYSIS AND TRAJECTORY REPORT

Flights RA-1 and RA-2

[redacted]

Contract No. AF 04(647)-592

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Lockheed Aircraft Corp., Sunnyvale, Calif. Missile and Space Div.

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LOCKHEED AIRCRAFT CORPORATION

MISSILES and SPACE DIVISION

FOREWORD

This Range Safety Analysis, including the separate Addendum Appendix containing the trajectory information, has been prepared by the Lockheed Missiles and Space Company under Contract AF 04(647)-592. Authority for preparation is contained in AFSSD Exhibit 58-1 and LMSC NASA/Agena B Program Work Statement LMSD-446432.

This report covers only the first two Ranger flights to be launched from Complex 12, Cape Canaveral, Florida and is submitted to comply with the AFMTC requirement that all agencies using the range facilities for missile flight operation must furnish data substantiating the Range Safety capabilities of the flight articles.

Trajectory data printout listings from the IBM 7090 computer are issued as a separate Addendum Appendix, with distribution limited on the basis of use requirement and need-to-know.

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SECTION 1
INTRODUCTION

1.1 GENERAL

The discussion and data in this report are required by the AFMTC Range Safety Officer (RSO) in establishing safe operating limits and destruct criteria for Ranger flights RA-1 and RA-2.

The purpose of these two flights is to obtain engineering test data and operating experience with the spacecraft vehicle system. The primary flight objectives are to establish the suitability of the system for subsequent hard-landing of an instrument package on the moon and secondarily to obtain scientific data on the space medium. A lunar trajectory is not flown on these first two system tests.

The composite vehicle will be launched from Complex 12 at AFMTC at a pad azimuth of 105 degrees. Shortly after liftoff, the Atlas will roll to the required trajectory of 108 degrees east of true north. The Agena first burn will inject the vehicle and spacecraft into a nearly circular parking orbit about the earth and Agena 2nd burn will inject the spacecraft into a highly elliptical orbit. The trajectories are such that all impulse phases will be monitored. from either land-based or shipboard tracking and telemetry stations.

The modified SM-65D Atlas missile will launch the composite vehicle and carry it through the ascent trajectory to a predetermined position and altitude, according to the guidance equations, where Atlas/Agena B separation will occur. The Atlas/Agena B separation sequence begins with jettisoning of the spacecraft shroud prior to separation, followed by separation. Agena is then pitched down to clear the shroud. After completion of the pitchdown,

Agena first burn ignition is initiated by the primary D-Timer. Agena first burn cutoff is commanded by the velocity meter when circular parking orbit conditions are attained. The Agena/spacecraft remains in this orbit until the point is reached where the Agena second burn is necessary to transfer the Agena and spacecraft into the final orbit. Agena second ignition is commanded by the primary D-Timer which is started by the auxiliary D-Timer. The auxiliary D-Timer is activated by a discrete command from the GE/ Burroughs Mod III-G guidance equipment during the Atlas ascent phase of flight.

Second burn cutoff is initiated by the velocity meter after velocity required to inject the spacecraft into an extended elliptical orbit has been achieved. Separation of the spacecraft is followed by an Agena B retro maneuver which will prevent it from following in the spacecraft trajectory. The unpowered Ranger spacecraft will continue on to complete it's scientific mission.

1.2 RANGE SAFETY REGULATIONS AND REQUIREMENTS

Since there are inherent dangers to life and property in the flight testing of space vehicles, it is mandatory that all reasonable precautions be taken to minimize these dangers. The Range Safety Program is designed to prevent accidents and incidents that may result in damage to property or injury to personnel from such operations.

Certain information and data are required by the Range in order to define safety limits and implement range safety procedures. This includes tabulations as a function of time for the predicted location of the vehicle; the instantaneous impact points in vacuum for nominal and three sigma deviation trajectories; and impact dispersions and maximum turn rate information. Plots are required of the estimated drag coefficients for re-entry bodies and re-entry pieces. Discussion is desirable on typical malfunctions and expected behavior of vehicle, expected effect of a destruct action, and other characteristics and/or peculiarities of significance.

The purpose of this report is to provide the needed range safety information on behalf of the launching agency.

1.3 DESCRIPTION OF VEHICLES

The composite flight configuration for all Ranger missions consists of three vehicles. These are the Atlas Series D modified, the Agena B modified, and the Ranger Spacecraft. The two booster vehicles both have several distinct impulse phases. In the Atlas they are the booster, sustainer, and vernier phases. In the Agena B the phases are first burn and second burn. The composite flight vehicle is shown in Figure A-1.

1.3.1 Atlas

The first booster is an Atlas SM-65D modified, Serial Nos. 111D and 117D. Propellants used are liquid oxygen and RJ-1. Flight control in pitch, yaw, and roll is accomplished by positioning the gimballed engines. Roll is controlled by the booster and vernier engines during the first phase and by the vernier engines during the last two impulse phases. Guidance is provided by the GE Mod III-G system. This radio-inertial guidance system consists of a rate beacon, pulse beacon, decoder, and associated X-band waveguide and antenna assemblies. The guidance system operates in conjunction with related ground equipment to perform rate measuring, position measuring, and data link functions. Position and rate information are compared with pre-computed stored trajectory information by a ground computer which generates command correction data. This data is received by the pulse beacon where it is decoded and superimposed on the gyro-inertial and programmer signals feeding the autopilot. The autopilot controls the engine thrust chambers to guide the missile onto the desired flight path, thus closing the guidance loop.

Requirements for the second booster and the Lunar mission necessitates certain structural and equipment modification in the Atlas booster. Major

differences between a Series D R&D Atlas and a modified Atlas for the Ranger program are as follows:

a. The forward liquid oxygen tank section skin thickness has been increased for added rigidity, necessitated by the weight of the Agena B and the spacecraft.

b. The standard re-entry vehicle adapter has been deleted and replaced with the Agena B adapter furnished by LMSD. This adapter incorporates a self-contained Agena B separation system.

c. The Atlas retrorockets, mountings and harness, have been deleted from the B-1 equipment pod and relocated to the Agena adapter.

d. A section of the B-1 pod-to-forward adapter wire way has been replaced by a larger streamlined housing and mounting provision has been made (at Station 675) for the forward rate gyros.

e. The Atlas vehicle equipment pods have been modified to reflect electronic equipment changes and to reduce overall weight. The upper pod is a modified B-1 pod similar to that used on the Project Mercury Atlas booster. The lower pod is extended to accommodate GE guidance equipment relocated from the upper pod.

f. The flight control system utilizes a more reliable transistorized autopilot, relocated rate gyros, and a modified programmer. The modified items in the programmer are the pitch program and switching junction.

g. The transistorized Mod III-G radio inertial guidance set replaced the Mod III-A because of weight and reliability considerations.

h. The Azusa system Type C transponder replaced the heavier Type B coherent transponder.

1.3.2 Agena B

The orbital booster is a modified Agena B, Model 10205, Serial Nos. 6001 and 6002. It has an engine restart capability and uses IRFNA and UDMH as propellants. Flight control is accomplished by two means after separation from the Atlas. Pitch and yaw are controlled by positioning the gimballed rocket engine during impulse phases and by cold gas jets during all other phases. Roll is controlled by the cold gas system at all times. The guidance system consists of an inertial reference package (IRP), a velocity

meter, a horizon sensor, sequence timers, attitude programming circuitry, attendant electronic assemblies, and a secondary junction box containing the telemetry conditioning circuits. This system requires signals from the Atlas booster to start the timer(s) and uncage the inertial reference package prior to separation from the Atlas. Following separation, this subsystem takes over and provides the guidance necessary to establish the desired trajectory.

1.3.3 Ranger Spacecraft

The Ranger spacecraft, Figure B-1, has been developed by the Jet Propulsion Laboratory specifically for the Lunar missions. RA-1 and RA-2 do not contain a propulsion system. After injection, maneuvers are limited to attitude control by means of cold gas jets. The spacecraft consists of the following components; vehicle structure, superstructure with omni-antenna, electrostatic analyzer, solar panel assemblies, 4-ft parabolic antenna, battery packs, and six modules containing various system components and scientific experiments. A protective shroud covers the payload during the Atlas boost phase. Components having the highest density and structural integrity are the six modules and battery pack. A weight breakdown of spacecraft modules appears in Table 1-1.

Table 1-1

RANGER SPACECRAFT HIGH DENSITY COMPONENT MODULES

Principal Components	Approximate Weight
Box one including 28-lb battery	60 lbs
Box two	37
Box three	35
Box four	28
Box five	36
Box six	41
Structure including 121-lb battery	232
Spacecraft total	671 lbs
Contingency	15 lbs

SECTION 2
GENERAL RANGE SAFETY PROBLEMS

2.1 GENERAL

In general the range safety problem is restricted to that portion of the flight that occurs between first motion and orbital velocity. Once orbital velocity has been achieved and thrust has terminated, the instantaneous impact point (IIP) leaves the surface of the earth and becomes indeterminate. The potential hazard to any area overflown by an orbiting vehicle then theoretically becomes nil. Therefore, in terms of range safety it is not necessary to consider those portions of flight that occur after the IIP becomes indeterminate.

Recent studies have shown that survival of hardware is possible after orbital decay and therefore a re-entry vehicle can become a safety problem. It is difficult, however, to predict the point of impact once an object has achieved orbital velocity. While certain malfunctions could prevent second ignition and cause such a problem, it is not considered to be within the range safety domain. Re-entry of the Agena B after second burn is also in this category.

It is conceivable, but most improbable, that the flight vehicle could assume an attitude such that the second burn impulse would cause premature re-entry. An extended first burn would cause an elliptical orbit if the flight vehicle attitude has not deviated excessively from the programmed values. Re-entry will result if the attitude has deviated excessively.

Re-entry of the flight vehicle due to a malfunction after Atlas sustainer cutoff would be expected to cause breakup similar to that of a destruct action. The extent of breakup and the dispersion of the re-entry pieces would be a function of the tangential velocity at the time of malfunction. The Range Safety Analysis for Ranger* considers two velocity conditions during the

* *Siegler Corporation, Inglewood, Calif., SSTG TR 61-48, NASA Ranger Program Range Safety Analysis, 10 March 1961, Secret*

Agena B first burn phase. They are: first, a premature thrust termination just before orbital velocity is reached, and second, a premature thrust termination a few seconds after first Agena B ignition. The first condition will generate maximum re-entry velocity and minimum re-entry angle while the second condition gives minimum re-entry velocity and a large re-entry angle. Large W/C_dA pieces were found to impact relatively close to the vacuum impact point, while small W/C_dA pieces were found to fall farther back-range. For the first condition, the impact point for the piece going the farthest distance was approximately 58 nm short of the vacuum impact point while those going the shortest distance impact approximately 165 nm short of vacuum impact point. This gives a dispersion of approximately 107 nm in an 11,000 foot corridor cross range. The dispersion for the second condition was approximately 30 nm.

The mean point of impact is influenced by three factors. They are the initial position of the vehicle at the time of the malfunction, the nature of the malfunction, and the prevailing wind during re-entry. The previously cited analysis considers these factors along with the lethal area of the re-entry pieces, the reliability of the flight articles, and the population and geography of the areas hazarded by overflight. This analysis computes the impact probability (P_I) and the kill probability (P_K) for three exit azimuths which have a flight altitude similar (93.7 nm vs 100 nm) to the RA-1 and RA-2 flights. These probabilities for the 108 degree azimuth are tabulated below. They are considered to be valid for the 100 nm altitude. The islands hazarded by overflight on this azimuth are Rochedas Sao Pedro e Sao Paulo, Ascension, and St. Helena. Probabilities were also computed as special cases for Antigua on a 108 degree azimuth, and for South America on a 114 degree azimuth. Since the numbers derived for these special cases are insignificant when compared with the numbers tabulated below, they are not included.

Table 2-1

SUMMARY OF P_I AND P_K FOR A 108 DEGREE EXIT AZIMUTH

	$P_I \times 10^{-5}$	$P_K \times 10^{-10}$
Africa	5.98	1,043.29
Islands	0.45	0.91
Total	6.43	1,044.20

During the period from shortly after launch to Atlas sustainer cutoff, the flight vehicle will not hazard a land area on the 108 degree azimuth since the IIP occurs on water. Moreover, trajectory deviations considerably in excess of those expected will not endanger a land area for the same reason.

Both the first and second boosters have a destruct capability through the Atlas Range Safety Command subsystem up to the point of Agena separation. In the event of an abnormal flight, there are no conditions other than the Range Safety Destruct Criteria for which a premature termination of flight is acceptable. This ensures receipt of the maximum possible data in order to determine the cause of such abnormal performance.

During launch the pad and immediate environs are endangered until the IIP moves over water. In the event of a malfunction this IIP can endanger land areas uprange and crossrange as well. Destruct action during this launch period can be as precarious as allowing impact on land. Past experience has shown that the destruct and propellant explosion effectively destroys and disperses most of the structural assembly. The larger pump assemblies and electronic packages that survived impacted in a direction of the planned trajectory, toward the ocean and away from the launch area.

2.2 FLIGHT ANALYSIS REQUIREMENTS

The Air Force Range Safety Division AFMTC has specified the contractor data submittal requirements for range safety analysis in a general set of instructions*.

The significant criteria in so much as these flights are concerned is item two of the instructions which states that trajectory data need only be presented from launch up to a point in flight where the remaining missile stage contains no emergency flight termination system. Therefore Appendices C and D contains data up to Agena separation.

2.3 WAIVER REQUIREMENTS

The Flight Range Safety Waiver requested in accordance with AFMTC instructions* is required because:

a. The instantaneous impact point will have to cross South Africa.

b. Due to weight considerations which would compromise the mission, a separate and independent destruct system with R-F receivers and attendant electronic equipment is not desired.

c. Trajectory azimuths may approach or exceed normal range safety limits.

This document, along with the impact probability study submitted previously**, is intended to support the waiver request mentioned above. The waiver of the Agena destruct system is based on the kill probability calculations of reference (c),** therefore an attempt to analyze all re-entry burn-up conditions is not attempted in this report.

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* *AFMTC/MTRSM/R.M.Montgomery/6517 (14pp), 9 June 1960*

** *SSTG TR 61-48, NASA Ranger Program Range Safety Analysis, Siegler Corp., Inglewood, Calif., 10 March 1961, Secret*

SECTION 3
MALFUNCTIONS

3.1 GENERAL

Most malfunctions are incorrect performance of the Guidance and Control or Propulsion subsystems. Failure to separate correctly or a tracking beacon failure are other malfunctions. A structural failure is most unlikely if the flight vehicle is performing normally. Abnormal performance in any one subsystem or a combination of subsystem malfunctions, can result in a structural failure.

Breakup will certainly result if the flight vehicle does not enter orbit. Factors contributing to vehicle breakup will vary, depending on the phase of flight. The most obvious, and least likely, is impact with either land or water caused by a malfunction during the early ascent phase. Control system malfunctions which are beyond compensation by the guidance system can cause bending failure during this period. Bending failure can also occur during certain periods with all power off. Tension failure can result under certain conditions when the vehicle tumbles. Aerodynamic forces will cause breakup if the allowable angle of attack is exceeded. Aerodynamic forces will also cause breakup during re-entry if the velocity is high enough. A destruct action can of course cause breakup at any time that this capability exists (launch to sustainer engine cutoff).

The malfunctions in each subsystem can be classified into major and minor categories. From a range safety point of view a "major" malfunction is one that will cause the flight vehicle to violate range safety restrictions while a "minor" malfunction prevents the flight vehicle from achieving normal trajectory.

The discussion that follows is indicative of the wide variety of failures which could be encountered during a flight. It is general in nature and is not intended to be exhaustive.

3.2 ATLAS FAILURE MODES

3.2.1 Minor Failures

3.2.1.1 Incorrect Pitch Program. A pitching rate that was too low would result in an abnormally high trajectory. In the extreme case of total absence of pitching, the missile would continue on a vertical flight path to engine burnout or cutoff. A high pitching rate would cause the vehicle to fly at a lower altitude than desired, and in the extreme case would continue to pitch downward until impact while still under propulsion. Pitching 180 degrees out of phase is another case that could occur. If the vehicle pitched 180 degrees out of phase from that which was programmed, it would start up-range over the mainland.

Any of the forementioned cases could cause the vehicle to assume a dangerous position or attitude. An incorrect pitch program could be caused by a variety of conditions such as incorrect wiring, fluctuation of the power supply, incorrect setting of the programmer pitch cam, and out-of-tolerance gyro off-set.

3.2.1.2 Incorrect Roll Program. The absence of programmed roll through the vertical portion of the trajectory immediately after launch would result in a trajectory azimuth of 105 degrees since the vehicle will be oriented on the launcher for flight in that direction. Minor errors in the roll program are corrected by the guidance system after booster cutoff.

Excessive roll of the vehicle during the vertical phase of flight would, generally, cause the vehicle to assume a course that could endanger overflown areas. If this type of irregularity could not be corrected immediately by the guidance system, the flight may have to be terminated.

The same type of malfunctions that would cause an incorrect pitch program could also cause an incorrect roll program. Both types of flight irregularities could occur during the same period, but this is considered improbable because of the chain of events that would have to take place simultaneously.

Closed loop tests are conducted during the countdown. This will assure that the booster is operative as an integrated unit. In particular it will demonstrate the proper operation of the flight programmer, guidance, autopilot, and control systems.

3.2.1.3 Minor Control Failures. Spurious commands to the rocket gimballing mechanism, or a malfunction of the mechanism itself, could produce improper maneuvering. These excursions could range from a slow yaw or pitch turn to one at the maximum turning rate. The probability of a prolonged turn at the maximum rate seems very low since the amount of engine gimballing must be controlled to maintain such a turn. In addition, the allowable angle of attack must not be exceeded at the lower altitudes.

Generally, control failures will be of the major type since the deflection angle of the rocket engines is critical.

3.2.1.4 Minor Propulsion Failures. Minor propulsion malfunctions are but remotely possible. Most failures in the propulsion system are expected to be of a major nature.

3.2.2 Major Failures

The following are typical of major failures which could be encountered during flight.

3.2.2.1 Major Control Failure. A major control failure could result from spurious rocket deflection, power supply fluctuations, or anomalous dynamic effects which are beyond the capabilities of the control system to counteract due to the limitations on rocket engine deflections. The booster engine is

limited to a maximum possible deflection of 4.5 degrees, of which 2 degrees are required to handle gusts and other disturbances. The remaining 2.5 degrees are available to trim steady-state aerodynamic lift. As a result, the permissible steady-state angle of attack is restricted to less than 10 degrees during the period from 30 to 100 seconds of flight. This trimmable angle of attack could be exceeded in less than 0.5 second following a major control failure.

It is estimated that the vehicle would fail structurally before it had rotated through 90 degrees during the first 100 seconds of flight while still under full propulsion. This could occur in less than 2 seconds in the event of major control failure.

3.2.2.2 Major Propulsion Failure. Total loss of thrust during the first 30 seconds of flight would result in impact within or adjacent to the Cape. Tumbling can occur at nearly any time during the first 100 seconds of flight because the flight vehicle is aerodynamically unstable.

Total loss of thrust by only one of the Atlas booster thrust chambers could send the vehicle into rapid tumbling motions, depending upon the deflection of the functioning chamber at the time of the malfunction. The most likely failures which could cause engine shutdown are listed below:

a. Severe shift in mixture ratio — the purpose of the propellant utilization system is to maintain a desired mixture ratio between the propellants as they are expended.

b. Pump or gas generator system failure — either failure will deny propellants to the engine(s).

c. Pump cavitation—this condition will cause the pump to overspeed which will result in a pump failure. A sufficient combination of dynamic and static head on the pumps throughout the vehicle's trajectory has been provided and tested on previous flights to give a high degree of assurance that this will not occur.

The vernier engines provide roll control during main engine burn. In the event they shut down during powered flight, the vehicle could roll, which would result in a loss of pitch/yaw control. This loss would produce an

erratic flight. The flight history of the Atlas shows that the possibility of this malfunction is very improbable. Assurance of correct vernier engine operation is obtained prior to Atlas release.

3.2.2.3 Major Structural Failure. Major structural failures could be caused by hard-over engine deflections, or by a combination of dynamic gust loads on the airframe. Scheduled wind soundings prior to launch will provide wind profile data which are teletyped to Sunnyvale where they are processed by a computer. The computer output data will be analyzed to determine whether or not wind conditions permit a safe launch. The launch recommendation will be transmitted to LMSD, Hanger E, at Cape Canaveral. Additional unscheduled soundings will be taken and processed if the forecast for the areas of interest indicate that a questionable wind condition may exist.

A vehicle break-up study was made by Convair and three types of failures investigated in both the pitch and yaw planes were felt to cover the extreme possibilities. It was concluded from this study that up to approximately 120 seconds after launch, bending failure will occur under the maximum load conditions in the yaw plane with both booster thrust chambers locked in the extreme position $\delta = \pm 4.5$ degrees. In the pitch plane from approximately 100 to 115 seconds, the vehicle could fail in bending with all power off. Bending rather than tension failure will occur up to 120 seconds since the angles of attack at which failure in bending occur are less than 90 degrees, while a tension failure results from a condition in which the vehicle is tumbling, with thrust control lost, and has passed through a 90 degree angle of attack during its rotation.

A check was also made on the possibility of bulkhead failure due to increased tension loads. It was concluded that a tension failure in the skin would occur before the bulkheads failed because of the larger margin of safety for the bulkheads.

The Atlas test vehicles have a design structural limit of 10 g's, but it is felt that the skin could take up to 11.5 g's in axial tension before failure.

3.2.3 Success Probability

An estimated probability of success can be made on the basis of twelve Atlas space booster launches to date. Two were failures. The classification of two others remain uncertain in that the cause of the flight failure could be either the Atlas booster or the respective second booster. Two probabilities can be derived: an optimistic one by assuming the two questionable flights to be successful, and a pessimistic one by assuming the two flights to be failures. The first probability will then be 5/6 and the second 2/3. One destruct action was taken in these twelve space booster flights. This is also the approximate destruct ratio for all Atlas flights.

3.3 AGENA FAILURE MODES

Due to the changing modes of vehicle operation, the effect of a subsystem or component malfunction varies greatly with the moment of occurrence. This discussion of Agena failure modes is arranged in the same order as the sequence of events.

The activation of guidance and attitude control in the Agena B/Ranger begins at Atlas/Agena B separation; however, certain events prior to this have a strong bearing on the behavior of the Agena B/Ranger after separation. Malfunctions before separation in the Atlas D portion of the flight vehicle have only indirect influence on this study and will be discussed separately. Relative probabilities of the most likely causes of failure in the Agena B are given in Table 3-1.

3.3.1 Failures Before Atlas/Agena B Separation

3.3.1.1 Primary D-Timer. The first start signal for the primary D timer is provided by an Atlas guidance command at 287.3 seconds followed by two

Table 3-1

AGENA B FAILURE PROBABILITIES

Circumstances	Probability
At first Agena engine start:	
Engine ignition failure	0.017723
Vehicle attitude control failure	0.004119
Loss of yaw control	0.000011
Vehicle reference heading incorrect	0.000002
Incorrect pitch attitude	0.000013
During first Agena engine burn:	
Premature engine cutoff	0.017700
Vehicle loses pitch or yaw control	0.000082
Engine locks hard over	0.000088
Engine locks near zero gimbal angle	0.000028

backup signals. Failure to start the timer will result in no operation of the Agena B/Ranger after Atlas/Agena B separation. Due to the design of the timer, only the following two failure modes should be recognized:

a. Synchronous motor and gear train stops.
b. Individual switch unit fails.

For item b, the consequences of the respective switch unit failures will be discussed at the point that the unit was scheduled to operate.

3.3.1.2 Uncage Agena Gyros. The Atlas VECO signal at 295.3 seconds also provides the "uncage gyro" command with two backup signals following. Failure to uncage any one gyro will result in loss of control of the corresponding function (pitch, yaw, or roll) after Atlas/Agena B separation.

3.3.1.3 Jettison Nose Fairing. Atlas guidance sends the first nose shroud ejection signal at 296.8 seconds followed by two backup signals. Failure to jettison the 181 pound nose shroud will not prevent the programmed attitude

and propulsion events to bring the Agena B/Ranger into a parking orbit but will prevent the Agena from achieving the desired final energy.

3.3.2 Failure During Atlas/Agena B Separation

The original command given by Atlas guidance is followed by two backup signals. Since the various errors and events at separation will continue to influence the Agena B/Ranger behavior after separation, these will be dealt with in detail.

3.3.2.1 Mechanical Separation. The Agena moves out of the Atlas-mounted adapter on dual rollers engaging three guide rails spaced symmetrically inside the adapter. Considerable clearance exists between rollers and rails. After firing the separation pin-pullers, a 200-pound thrust is required to initiate the separation movement, and a 35-pound thrust is needed to sustain the motion. At 299.8 seconds, Atlas guidance commands the start of the separation cycle (followed by two backup signals). This fires the pin-puller squibs and ignites two adapter-mounted retrorockets. Each of the latter develops sufficient thrust to execute separation. A separation switch will then activate the pneumatic attitude control system. Should the retrorockets fail to ignite, the subsequent firing of the two Agena ullage rockets provides enough thrust to push the Agena B out of the adapter.

Failure of the pin-pullers to release will not only retain the mechanical bond between Atlas and Agena but will also prevent the attitude control system from operating. The flight vehicle will retain the same relative attitude as it had at Atlas VECO, subject to the angular and linear velocities prevailing at VECO. The primary D-timer will continue to operate, as the separation switch has no "hold-fire" circuit, and will fire the two ullage rockets at 336.3 seconds.

It is expected that the ullage rockets, being pointed outwards, will burn holes in the adapter wall but not cause any other damage. The Agena engine start signal will come from the D-timer at 348.3 seconds and a normal

start sequence is anticipated. As the Agena engine thrust is not sufficient to overcome the holddown bolts, it is probable that the engine jet will destroy the front end of the Atlas and free the adapter. If an explosion damaging the Agena does not occur, a fly-away will take place, but the roll attitude control system will be still inoperative. The 9 degree Agena pitch-over program will cause the engine to go hard over to correct the position; however, the center of gravity will be behind the engine gimbal point and a further pitch-up will occur until the Atlas falls away.

3.3.2.2 IRP Gyro Saturation. The three gyros used for pitch, yaw, and roll control will saturate at a 12 degree error. If only one retrorocket fires, a pitch rate of 0.12 degree per second will be imparted to the Agena and the time to completely separate from the Atlas will take 6 instead of 3 seconds. During separation, experience with previous flights indicate a maximum Agena B turn rate of 1 degree per second as a result of mechanical "bumping". Both these values will cause Agena gyro errors well below the permissible limits for the gyros used.

3.3.3 Failures After Atlas/Agena B Separation

3.3.3.1 Coast Period. When separation is completed at about 301.8 seconds, the Agena B/Ranger coasts for 34.5 seconds. After 11.5 seconds, a high rate pitch-down program is initiated, lasting three seconds. A lower pitch-down program is then begun and at the same time the horizon sensor (H/S) is connected to the pitch and roll gyros. This latter program continues through the first Agena engine burning period.

3.3.3.2 Time Period 301.8 to 313.3 Seconds. At separation, the pneumatic attitude control system is activated and the IRP, through the flight control electronics and the gas jets, retains the vehicle attitude at VECO. The following failure modes are possible:

a. Excessive or lack or yaw control
b. Excessive or lack of pitch control
c. Excessive or lack of roll control.

As the saturation rate of turn for the gyros in the IRP package will be rapidly attained if any one attitude control jet opens and stays open (0.4 sec for roll, 2.8 sec for pitch and yaw gyro), the vehicle will be out of control. The same applies if complete lack of control should occur, while a reduced rate of control will not necessarily affect the possibility of entering the parking orbit.

3.3.3.3 Time Period 313.3 to 316.3 Seconds. At 313.3 seconds, the primary D-timer commands a high rate (180 degree/minute) pitch-down movement while retaining yaw and roll attitude as before. This will enable the Agena B/Ranger to pitch down 9 degrees to the proper attitude for first Agena engine burning. In addition to the failure modes of the previous period, the following mode is possible.

If the primary D-timer does not command high rate pitch-down the comments of Paragraph 3.3.3.2 are applicable.

It should also be noted that a high pitch-up attitude error will remain at the time of the Agena engine's first start, if the primary D-timer does not command a high rate pitch-down, since the horizon sensor (H/S) will not have sufficient gain to restore the vehicle to near horizontal attitude during the time period 316.3 to 348.3 seconds.

3.3.3.4 Time Period 316.3 to 336.3 Seconds. At 316.3 seconds, the primary D-timer switches the high rate pitch-down program to a lower rate pitch-down (3.54 degree/minute) and connects the H/S to the pitch and roll gyros with a control level of 3 degree/minute/degree of angular error. The combined output of the commanded pitch-down rate and the H/S program is calculated to keep the Agena B/Ranger at an average pitch attitude of +3.2 degrees above local horizon. In addition to the failures listed in Paragraph 3.3.3.2, the following modes are possible:

a. Primary D-timer does not change from high to low pitch-down rate

b. Horizon sensor is not connected to IRP gyros.

Failure of changeover to the lower pitch-down rate will cause the vehicle to continue over at approximately 3 degree/second. Should the H/S fail to give input to the gyros, the resulting pitch-plane control would give too low a vehicle attitude during the early period of Agena engine burning and too high an attitude at the end of the period.

It should be recognized that a failure of certain parts in the IRP and flight control electronics could conceivably cause temporary saturation of the yaw gyro by creating an excessive turn rate. This would force the gyro to take up a changed reference heading. Such an intermittent failure would then result in the vehicle taking up a new flight azimuth. The probability of such an event is covered in the following paragraphs.

3.3.3.5 First Agena B Engine Burning. Agena first burn is scheduled for the period 336.3 to 498.9 seconds. This time period also includes a portion of the ullage rocket burn time which ensures engine ignition. At 336.3 seconds, the velocity meter is activated, and the ullage rockets ignited. At 348.3 seconds the main Agena engine start cycle is originated by the primary D-timer which also shifts pitch and yaw control from pneumatic to hydraulic (engine gimballing) operation. The pitch program is unchanged and roll control is still executed by the IRP package through the gas jets. As the engine cutoff circuit is not activated until 487.3 seconds, the engine should not stop until after that time. The velocity meter cutoff point is nominally scheduled for 498.9 seconds. The backup signal for engine cutoff is sent by the primary D-timer at 507.3 seconds.

The major changeover of control functions at the beginning of this period justifies tabulating the applicable failure modes during the first Agena engine run:

a. Engine fails to start
b. Premature engine cutoff
c. Excessive or lack of roll control (pneumatic)
d. Excessive or lack of yaw control (hydraulic)

e. Excessive or lack of pitch control (hydraulic)

f. H/S failure

g. Primary D-timer fails to enable engine cutoff, or send backup stop signal

h. Velocity meter provides early or late cutoff signal.

For this study, engine starting failure will result in no safety complications because the IIP is on water. An early engine cutoff could result in undesirable premature downrange impacts, while the difficulties in roll, yaw, and pitch control would result in loss of control. A failure of the primary D-timer to activate engine cutoff would lead to engine burning until fuel depletion, which in turn would result in too high an orbital velocity and an elliptic parking orbit. As the primary D-timer prevents the velocity meter from sending an early engine cutoff signal before 487.3 seconds, near parking orbit velocity would be achieved. Engine cutoff at 487.25 seconds will, for nominal and 3σ trajectories, result in Agena impacts as given in Table 3-2. Impact occurs in a safe area in both nominal and 3σ trajectories. Trajectory parameters for a time corresponding to the "backup" engine cutoff setting, as given in Table 3-3, are orbital for both the nominal and 3σ slow trajectories. Late engine cutoff (507.3 seconds) would provide an orbital velocity only slightly in excess of the desired one.

3.3.3.6 Structural Failure During Ascent. Separation of the Agena B from the Atlas during flight would be the result of a failure of the tension ties at Station 388.55; or, a bending failure in other areas, namely, Agena Stations 377 to 388, and in the Atlas at approximately Station 850. The most probable cause of such a failure would be that one or more engines were locked hard over. Separation of Station 388.55 would automatically fire the premature separation destruct circuit, destroying the Agena B.

Table 3-2

IMPACT DATA FOR PREMATURE FIRST BURN CUTOFF

	TC (sec)	Latitude (deg)	Longitude (deg)	TF (sec)	Range (nm)	Downrange (ft)	Crossrange (ft)
Nominal Trajectory	487.25	-6.9103063	17.904608	840.07531	4193.4333	35437701	663446.99
3σ Fast Trajectory	487.25	-15.139989	4.9457819	1065.4061	5101.1126	30783476	716739.35

Table 3-3

TRAJECTORY PARAMETERS AT TIME OF ENGINE CUTOFF
BACKUP SIGNAL

	Altitude (ft)	Inertial Velocity (ft/sec)	Inertial Flight Path Angle (deg)
Nominal Trajectory	611354.77	26191.429	.09126363
3σ Slow Trajectory	607049.17	25714.483	-.10874584

SECTION 4
FLIGHT SAFETY CRITERIA DATA

4.1 GENERAL

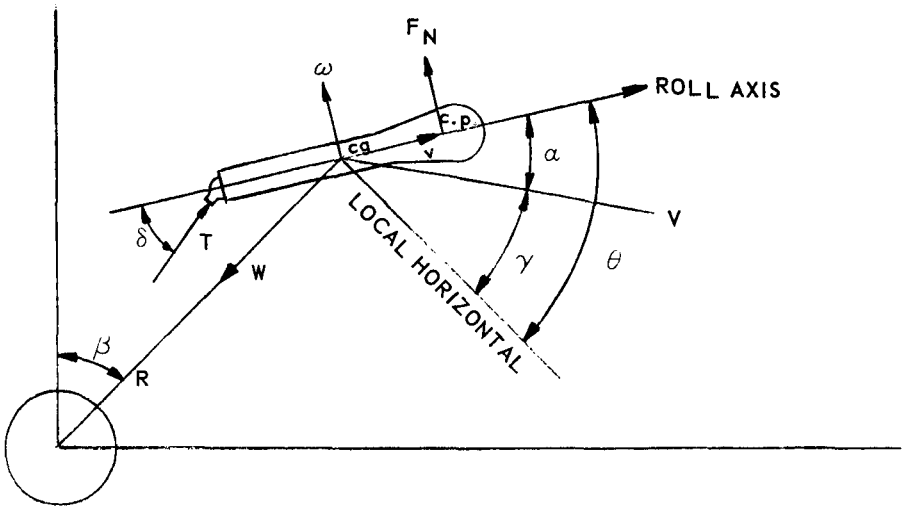
Certain characteristics and expected behaviors of the flight vehicle are discussed in this section. Each discussion item is included to fulfill some requirement not otherwise covered in another part of this report. They supplement the tabulations contained in the appendices as data to establish a safe flight criteria.

4.2 TUMBLING TURN RATE DATA

Tumbling turn rate data for the Atlas D have been calculated* by CV-A for a representative flight vehicle. Calculations for tumbling turns have also been made at LMSD for a flight vehicle very similar to the Ranger vehicle. The results of these LMSD calculations have shown that the flight path angle magnitudes are either equal to or somewhat less than those given in the referenced CV-A document; and this data may therefore be used without modification for Ranger flights RA-1 and RA-2.

It is recognized that this could result in a somewhat more restrictive destruct criteria than if actual rates were used. The method of calculation of a tumbling vehicle is given below. Typical results from the LMSD calculations are compared in Figure 4-1 with the turn rates contained in CV-A calculations*.

* *Convair-Astronautics, AZN-27-117, Angular Turn of Velocity Vector Due to Tumbling D Series Missile, 15 June 1959, Confidential*



4.2.1 Method of Calculation of Tumbling Vehicle

A tumbling vehicle under the action of gravity, aerodynamics, and thrust is shown. The roll axis of the missile makes an angle θ with the local horizontal. The angular velocity of the roll axis is given by

$$\dot{\theta} = \dot{\beta} + \dot{Q} \tag{1}$$

where $\dot{\beta}$ represents the rotation of the center of gravity about the earth's center and $\dot{Q}$ represents the rotation of the vehicle about its center of gravity. The first term of Equation (1) is given by the relation:

$$\dot{\beta} = \frac{V \cos \gamma}{R} \tag{2}$$

and the second term is given by the relation

$$\dot{Q} = \int \frac{1}{I_{zz}} \left[\dot{Q} C_{m\dot{Q}} q S + F_N \overline{X} + T \sin \delta \overline{X}_T + \frac{\dot{W}}{g_o} \overline{X}_e^2 \dot{Q} \right] \tag{3}$$

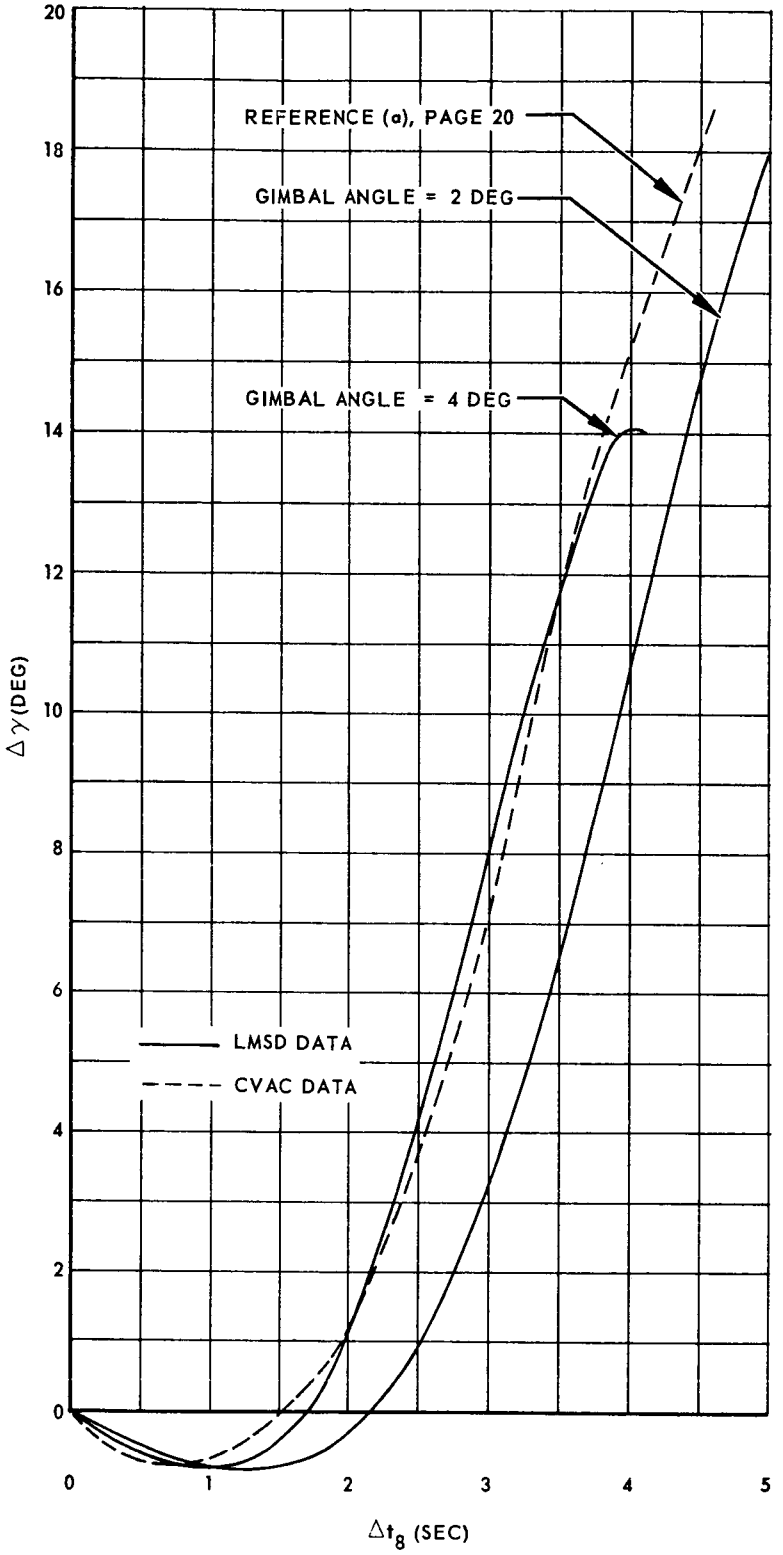


Figure 4-1 Turn Angle of Velocity Vector vs t for time = 24 seconds

where

V = velocity, ft/sec
T = thrust, lbs
δ = gimbal angle, deg
α = angle of attack, deg
γ = flight path angle, deg
F_N = aerodynamic normal force, lb
g_o = acceleration of gravity at earth's surface, ft/sec^2
I_{zz} = pitch moment of inertia, slug-ft^2
$\overline{X}$ = distance between CG and CP, ft
$\overline{X}_T$ = distance between CG and engine gimbal point, ft
$\overline{X}_e$ = distance between CG and nozzle exit plane, ft
$\dot{W}$ = fuel flow rate, lb/sec
$C_{m_{\dot{Q}}}$ = damping in pitch coefficient
S = reference area, ft^2
q = dynamic pressure, lbs/ft^2
R = radius from the center of the earth to the missile CG, ft

In order to evaluate aerodynamic coefficients in Equation (3), it is necessary to consider high angles of attack. The coefficients will be increased over those for the zero-lift condition due to the crossflow coefficients at high angles of attack. The equations for center of pressure location and aerodynamic normal force are:

$$X_{cp} = \frac{C_{N_B}(CP_o) + C_{N_C}(CP_c)}{C_{N_B} + C_{N_C}} \tag{4}$$

$$F_N = qS(C_{N_B} + C_{N_C}) \tag{5}$$

where

$C_{N_B} = C_{N_{\alpha_o}} \cos \alpha \sin \alpha$

$C_{N_{\alpha_o}}$ = zero lift normal force coefficient

CP_o = center of pressure at $\alpha = 0$ deg

$C_{N_C} = \frac{S_B}{S_{ref}} \sin^2 \alpha C_{D_\infty}$

$S_{B/S_{ref}}$ = ratio of planform area to reference area

C_{D_∞} = crossflow drag coefficient

CP_c = the station at which the crossflow normal force acts

All items in Equations (1) through (5) are instantaneous vehicle parameters. The solution of this set of equations gives the instantaneous turning rate in the vertical plane. Corresponding rates for the horizontal plane are obtained by adding the gravitational correction term:

$$\frac{g \cos \gamma}{V} \tag{6}$$

The change in θ over any given time interval is found by integration of Equation (1). The angle of attack, α, is given by the relation

$$\alpha = \tan^{-1} \frac{\omega}{\mu} \tag{7}$$

where

ω = component of missile velocity along yaw axis

μ = component of missile velocity along roll axis

Finally, the angle turned through by the velocity vector during the tumble is given as:

$$\Delta \gamma = \Delta \theta - \Delta \alpha \tag{8}$$

4.3 EXPECTED EFFECTS OF A DESTRUCT ACTION

4.3.1 General

It is expected that a destruct action will result in complete dispersion of the fuels and breakup of both booster vehicles. The extent of these results will depend on the time during flight at which the destruct action is taken. Some of the variables involved are the amount of fuel remaining in each tank, the velocity of the vehicle, and the density of the atmosphere. It is expected that nearly all the fuel will be consumed through either explosion or burning. The explosive force will depend on the quantity of fuel remaining and degree of mixing. The quantity of fuel remaining will not change for the Agena B booster for the duration of the command destruct capability. For the Atlas, fuel quantity will decrease nearly linearly during each impulse phase.

The velocity of the vehicle contributes to breakup while the vehicle is in denser atmosphere. The degree of breakup may be less if the destruct action is taken at a later time during Atlas impulse when the remaining fuel is less and the atmospheric density is lower. The vehicle velocity, however, will be higher and the re-entry conditions will be more severe. This will result in additional breakup on re-entry. In general, the extent of breakup will be proportional to the kinetic energy of the vehicle at the time of the destruct action.

4.3.2 Estimated Number and Size of Pieces

On the basis of these considerations it is expected that the size and quantity of pieces that will survive to surface impact will vary with the time at which the destruct action is taken. It is expected that a few large pieces will result and survive from an early destruct action and that only high density pieces having a greater structural strength are expected to survive for either a late destruct or a normal orbital decay re-entry. The effects of a destruct action at any other time will fall in between these extremes. The number of pieces will increase with time because of the energy contained in the vehicle, which

increases with time, and the reaction of the destructed vehicle with the atmosphere during either ascent or re-entry. The number of dangerous pieces however, will decrease with time for the same reason. An estimate of the surviving major pieces (exclusive of skin pieces, fragments, or molten metal pellets) and weights are tabulated below:

a. Early destruct

Surviving Piece	Weight (lb)
Atlas booster section	7200
Atlas external equipment pod	1600
Atlas external equipment pod	1800
Atlas/Agena adapter and Agena aft section	700
Agena battery	140
Ranger payload including adapter and shroud	800

b. Late destruct

Surviving Piece	Weight (lb)
Atlas thrust chamber gimbals	430
Atlas pump drive and turbine	230
Atlas vernier engines (each)	50
Agena forward equipment rack	400
Agena rocket motor	300
Ranger (several pieces)	20-80

One destruct action has been taken on an Agena booster where the pieces have been recovered. This was Discoverer X which was destroyed at 20,900 feet. The second booster of this composite vehicle was an Agena A. It is expected that a destruct action on an Agena B under the same conditions would be very similar. The dispersion of the pieces suggests that all breakup occurred prior to impact. The recovered pieces of the Discoverer X Agena A are listed below:

Engine
Forward section and Payload
Hydraulic package
Nitrogen sphere
Helium sphere
Several large skin pieces

4.3.3 Effects of Explosion on Velocity of Pieces

At all but the earliest phases of a vehicle flight, the forward velocity imparted to fragments by virtue of the vehicle velocity will be much greater than that caused by the explosive force. For flight times greater than 30 seconds, the forward velocity imparted to the pieces by the explosion can be considered to be negligible. The energy required to cause lateral dispersion of the pieces, normal to the vehicle velocity vector, will be supplied solely by the forces of the explosion. The initial lateral velocity imparted to an average size fragment by this force is estimated to be not more than 250 FPS.

4.4 DRAG ESTIMATE PLOTS

The expected effects of destruct explosion, estimates of C_d versus Mach number for pieces which, after destruct, are expected to travel maximum and minimum distances, and the effects of explosion on velocity of pieces for the Atlas vehicle are contained in reference (b).*

Calculations for the Agena stage have shown that the range of C_dA/W for the maximum and minimum distance pieces from the Agena vehicle lie within the C_dA/W range given in reference (b) for the Atlas vehicle.

4.5 MISCELLANEOUS DATA

The tracking equipment on board the flight vehicle which can be used for range safety purposes is as follows:

a. Atlas telemetry transmitter
b. Atlas Azusa transponder
c. General Electric Mod III-G transponder
d. Agena "C" band transponder

* *Convair Report ZC-7-136, subject "Trajectory and Aerodynamic Information for XSM-65 Missile Flights," 26 June 1959, Secret*

The approximate time delay between activation of mainstage firing circuits and first vehicle motion is 4.5 seconds. First motion is defined as two inches of movement.

The trajectory deviations expected for the most adverse wind condition for which a launch could be expected are contained in the 3σ trajectory tabulations. The effects of nonstandard variations in wind, yaw gyro drift rate, and roll gyro drift rate were included in these calculations.

The nominal duration of each impulse phase can be derived from the trajectory tabulations contained in the addenda to the appendices.

The nominal duration of Atlas vernier phase is 18.43 seconds.

The maximum burn time and maximum impact range are 307.6 seconds after launch and 1701 nm downrange, assuming the Atlas burns to fuel exhaustion.

SECTION 5
FLIGHT TERMINATION SYSTEM

5.1 GENERAL

Flight termination systems are provided on each booster of the Ranger vehicle. They provide for the termination of all flight capability from launch to Atlas sustainer engine cutoff. Since the spacecraft contains no propellants, a flight termination system is not considered necessary. Automatic termination is provided in the event of inadvertent separation between the first and second boosters. These systems include a safety feature which will preclude inadvertent detonation of the destruct charges during preflight checkout and countdown on the pad. They are as follows:

a. Each system can be armed and disarmed remotely from the blockhouse

b. Arming devices have a mechanical lockout to prevent erroneous arming. (A pin must be manually removed before either destruct package can be armed)

c. Simulators are substituted for the destruct package during prelaunch checkout.

5.2 ATLAS FLIGHT TERMINATION SYSTEM

This system is known as the Range Safety Command subsystem on the Atlas. It is a frequency modulated system which is compatible with the ground command destruct system at AFMTC. The system consists of two independent command destruct paths which have in common the antenna and coupler and the main destruct charge. Each path consists of the antenna and coupler, command receiver, battery, arming device, and primer and booster charges. The units of the command receiver are: broadband converter, variable IF amplifier, fixed IF amplifier and discriminator, audio amplifier, audio tone filters and detectors, channel relay amplifiers

and relays, and a transistorized power supply. The arming device consists of two relays and a switch which short-circuits the primer leads and isolates the destructor units from receiver commands when in a safe condition. The coupling arrangement between the antennas and the receivers is such that the radiation pattern seen by one receiver complements the radiation pattern seen by the other. As long as both receivers are operative, essentially omnidirectional coverage is provided. If one receiver should fail approximately 85 percent coverage is provided with an antenna gain that is greater than 10 db below that of an isotropic radiator. Either fuel cutoff or a destruct action will be effected when the appropriate tone combination is received by the system. . Detailed information on this system is provided in LMSD-447590-A, Ranger Flight Termination System, 28 February 1961, Confidential.

5.3 AGENA B FLIGHT TERMINATION SYSTEM

This system consists of two independent circuits which utilize the dual command destruct signal inputs from the Atlas first booster. These circuits and destruct charge are located in the adaptor section between the Agena B and the Atlas. This provision will permit the destruct system to remain with the Atlas on separation which will eliminate weight and destruct capability during the Agena boost phase.

Dual independent premature separation destruct circuits are also provided. Each circuit consists of independent battery, wiring, and squibs for the initiation primer. These circuits are actuated when a lanyard releases either of the two (2) separation switches mounted on the adaptor at the base of the vehicle. Some components of each circuit are shared with the command destruct circuits. These circuits are disarmed and "safed" before programmed separation of the Agena B. Detailed information on this system is provided in LMSD-447590-A, Ranger Flight Termination System, 28 February 1961, Confidential.

APPENDIX A

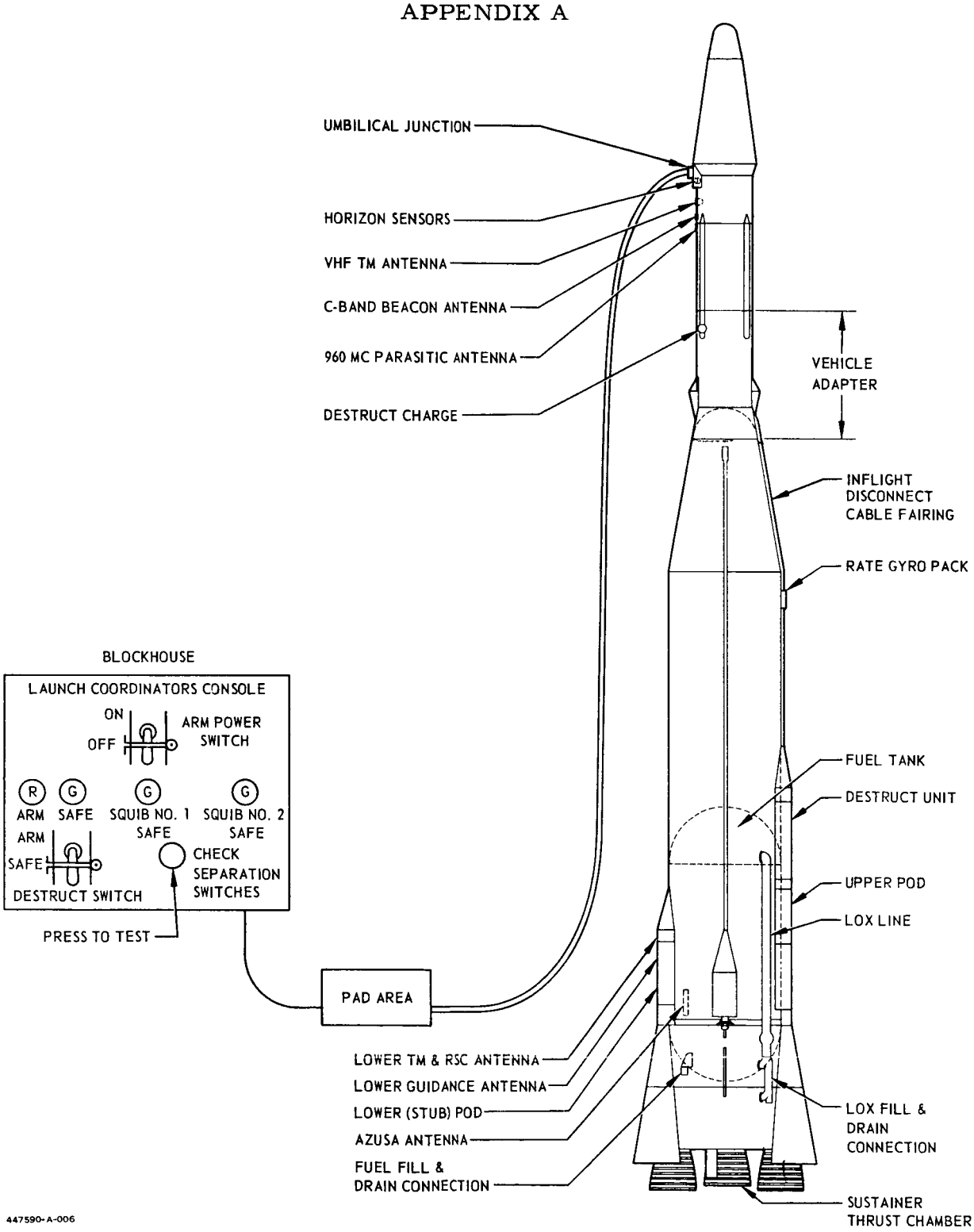


Figure A-1 Agena-B/Atlas Vehicle

APPENDIX B

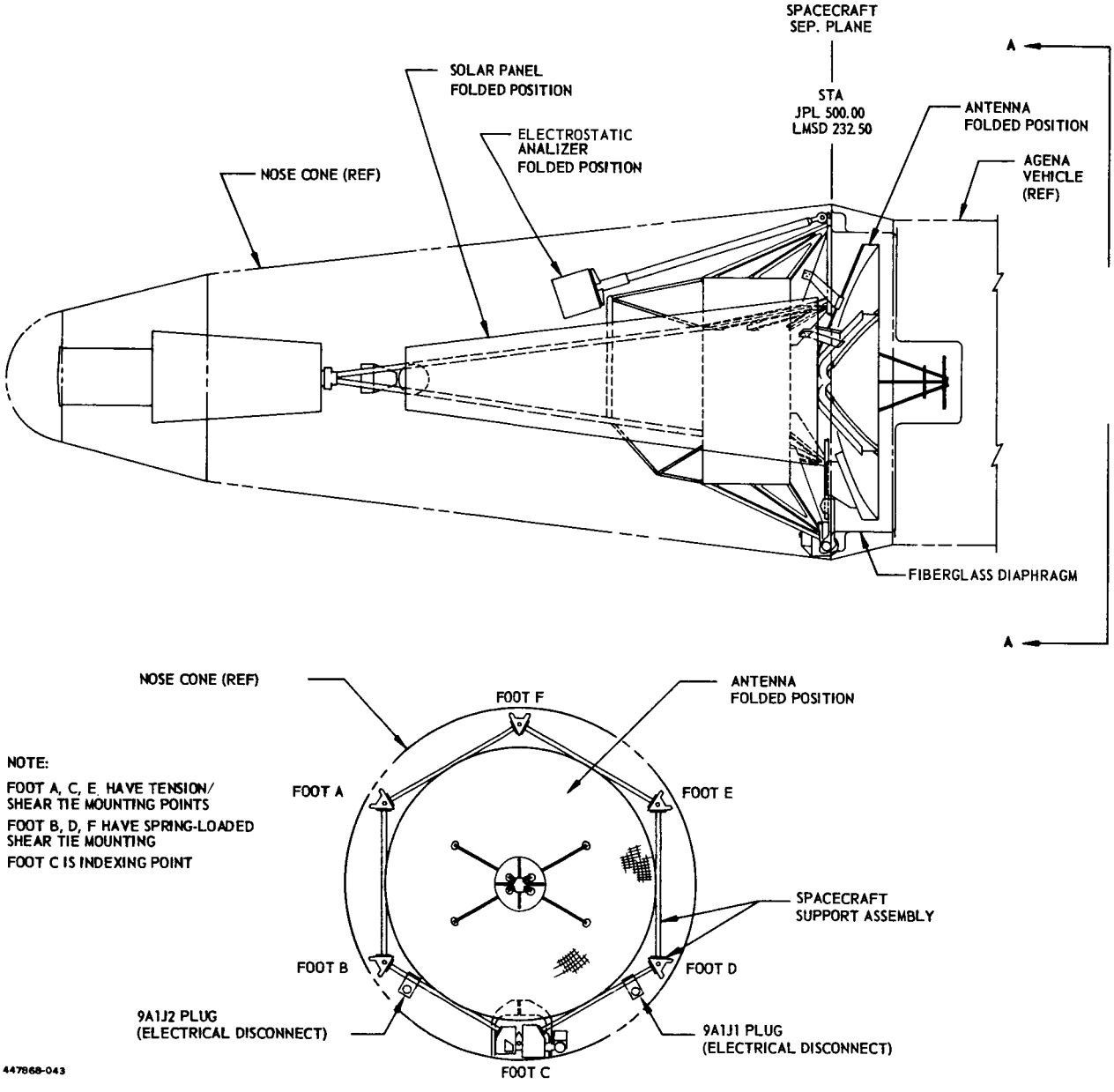


Figure B-1 Ranger Spacecraft, Basic Structure

APPENDIX C
VACUUM IMPACT POINTS PREMATURE ATLAS CUTOFF

C. 1 ADDENDUM APPENDIX TABLES

Trajectory data printout listings from the IBM 7090 computer are issued as a separate Addendum Appendix, with distribution limited to the list shown under "Distribution" at the end of this document.

Tables C-1 thru C-5 contain a locus of vacuum impact points for various premature cutoff times along the nominal, 3σ maximum, 3σ minimum, and 3σ lateral right trajectories generated using a closed-loop simulation incorporating the effects of the Atlas ascent guidance equations. These are the same trajectories contained in Appendix D, Tables D-1 thru D-4 and include times corresponding to booster cut-off, sustainer cut-off and vernier cut-off.

Table C-1 gives the expected impact points for the Atlas booster and sustainer packages if they do not burn up upon re-entry into the atmosphere. In the event burn up does not occur then data on the pieces will be comparable to that given in Reference a.

Tables C-2 thru C-4 give the expected impact dispersion points about the nominal for the Atlas booster and sustainer packages. Impact points are on an ellipsoidal, rotating earth.

Table C-5 contains the impact points for a normal Atlas going to fuel exhaustion which apply if normal thrust termination does not occur.

APPENDIX D
RANGER TRAJECTORY TABLES

D. 1 ADDENDUM APPENDIX TABLES

Trajectory data printout listings from the IBM 7090 computer are issued as a separate Addendum Appendix, with distribution limited to the list shown under "Distribution" at the end of this document.

Tables D-1 thru D-5 contain the computer listings simulating the Atlas phase of flight for the nominal trajectory; and, for the 3σ trajectories of maximum performance, minimum performance and lateral right deviation. The 3σ left deviation trajectory is assumed to be symmetrical to that of the 3σ right deviation. Tables D-1 thru D-3 contain the following data at one second intervals.

a. Position coordinates (x, y, z) (ft) vs time in sec.

b. Velocity components ($\dot{x}$, $\dot{y}$, $\dot{z}$) in ft/sec vs time (sec)

c. Speed $\sqrt{\dot{x}^2 + \dot{y}^2 + \dot{z}^2}$ (ft/sec) vs time (sec)

d. Path angle of velocity vector relative to the local horizontal (deg) vs time (sec)

e. Altitude above earth's surface (ft) vs time (sec)

f. Ground range from launch pad to a point directly beneath the missile (ft) vs time (sec)

g. Total weight (lbs) vs time (sec)

h. Thrust (lbs) vs time (sec)

i. Longitudinal acceleration (ft/sec^2) vs time (sec)

The position and velocity values are referenced to an orthogonal, earth-fixed coordinate system with origin at the launching pad. The nominal trajectory is based upon nominal weight and engine log thrust and fuel flow data. The 3σ maximum and 3σ minimum performance trajectories

define at any time after launch the limits of normality as far as impact range is concerned. The 3σ maximum-performance (fast, light, lofted missile) was assumed to have high thrust, low takeoff weight, low specific impulse, head winds, and reduced booster stage programmed pitch program. The 3σ minimum performance (slow, heavy, low missile) has low thrust, high take-off weight, high specific impulse, tail winds and increased booster stage programmed pitch program.

It should be noted that Table D-5 presents the above trajectory data for a normal Atlas going to fuel exhaustion, applies if normal thrust termination does not occur. All of the trajectory tables in the Addendum Appendix have been calculated using wind profile magnitudes, above which a launch would not be attempted and therefore give the maximum expected trajectory deviation during the first 60 to 70 seconds of flight.

REFERENCES

(a) AZN-27-117, Angular Turn of Velocity Vector Due to Tumbling D Series Missile, Convair-Astronautics, San Diego, Calif., 15 June 1959 [redacted]

(b) ZC-7-136, Trajectory and Aerodynamic Information for XSM-65 Missile Flights, Convair-Astronautics, San Diego, Calif., 26 June 1959 [redacted]

(c) SSTG TR61-148, NASA Ranger Program Range Safety Analysis, Siegler Corp., Inglewood, Calif., 10 March 1961 ([redacted])

(d) General instructions to distribution from AFMTC/MTRSM/RM Montgomery/6517 (14 pp), Trajectory and Aerodynamic Data Requirements for Atlas, Titan, Atlas-Able, Atlas-Mercury, and Similar Missiles, 9 June 196[redacted]

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